(ii) The value of HIC is defined as—

$$HIC = \left\{ (t_2 - t_1) \left[\frac{1}{(t_2 - t_1)} \int_{t_1}^{t_2} a(t)dt \right]^{2.5} \right\}_{Max}$$

Where—

t_1 is the initial integration time, expressed in seconds, t_2 is the final integration time, expressed in seconds, and a(t) is the total acceleration vs. time curve for the head strike expressed as a multiple of g (units of gravity).

(iii) Compliance with the HIC limit must be demonstrated by measuring the head impact during dynamic testing as prescribed in paragraphs (b)(1) and (b)(2) of this section or by a separate showing of compliance with the head injury criteria using test or analysis procedures.

(6) Loads in individual shoulder harness straps may not exceed 1,750 pounds. If dual straps are used for retaining the upper torso, the total strap loads may not exceed 2,000 pounds.

(7) The compression load measured between the pelvis and the lumbar spine of the ATD may not exceed 1,500 pounds.

(d) For all single-engine airplanes with a V_{SO} of more than 61 knots at maximum weight, and those multiengine airplanes of 6,000 pounds or less maximum weight with a V_{SO} of more than 61 knots at maximum weight that do not comply with § 23.67(a)(1);

(1) The ultimate load factors of § 23.561(b) must be increased by multiplying the load factors by the square of the ratio of the increased stall speed to 61 knots. The increased ultimate load factors need not exceed the values reached at a V_{S0} of 79 knots. The upward ultimate load factor for acrobatic category airplanes need not exceed 5.0g.

(2) The seat/restraint system test required by paragraph (b)(1) of this section must be conducted in accordance with the following criteria:

(i) The change in velocity may not be less than 31 feet per second.

(ii)(A) The peak deceleration (g_p) of 19g and 15g must be increased and multiplied by the square of the ratio of the increased stall speed to 61 knots:

g_p=19.0 $(V_{S0}/61)^2$ or g_p=15.0 $(V_{S0}/61)^2$

(B) The peak deceleration need not exceed the value reached at a V_{S0} of 79 knots.

(iii) The peak deceleration must occur in not more than time (t_r), which must be computed as follows:

$$t_r = \frac{31}{32.2(g_p)} = \frac{.96}{g_p}$$

where—

g_p=The peak deceleration calculated in accordance with paragraph (d)(2)(ii) of this section

t_r=The rise time (in seconds) to the peak deceleration.

(e) An alternate approach that achieves an equivalent, or greater, level of occupant protection to that required by this section may be used if substantiated on a rational basis.

[Amdt. 23–36, 53 FR 30812, Aug. 15, 1988, as amended by Amdt. 23–44, 58 FR 38639, July 19, 1993; Amdt. 23–50, 61 FR 5192, Feb. 9, 1996; Amdt. 23–62, 76 FR 75756, Dec. 2, 2011]

FATIGUE EVALUATION

§ 23.571 Metallic pressurized cabin structures.

For normal, utility, and acrobatic category airplanes, the strength, detail design, and fabrication of the metallic structure of the pressure cabin must be evaluated under one of the following:

(a) A fatigue strength investigation in which the structure is shown by tests, or by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected in service; or

(b) A fail safe strength investigation, in which it is shown by analysis, tests, or both that catastrophic failure of the structure is not probable after fatigue failure, or obvious partial failure, of a principal structural element, and that the remaining structures are able to withstand a static ultimate load factor of 75 percent of the limit load factor at $V_{C,}$ considering the combined effects of normal operating pressures, expected external aerodynamic pressures, and flight loads. These loads must be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise considered.

(c) The damage tolerance evaluation of § 23.573(b).

(d) If certification for operation above 41,000 feet is requested, a damage tolerance evaluation of the fuselage pressure boundary per § 23.573(b) must be conducted.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23–14, 38 FR 31821, Nov. 19, 1973; Amdt. 23–45, 58 FR 42163, Aug. 6, 1993; Amdt. 23–48, 61 FR 5147, Feb. 9, 1996; Amdt. 23–62, 76 FR 75756, Dec. 2, 2011]

§ 23.572 Metallic wing, empennage, and associated structures.

(a) For normal, utility, and acrobatic category airplanes, the strength, detail design, and fabrication of those parts of the airframe structure whose failure would be catastrophic must be evaluated under one of the following unless it is shown that the structure, operating stress level, materials and expected uses are comparable, from a fatigue standpoint, to a similar design that has had extensive satisfactory service experience:

(1) A fatigue strength investigation in which the structure is shown by tests, or by analysis supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected in service; or

(2) A fail-safe strength investigation in which it is shown by analysis, tests, or both, that catastrophic failure of the structure is not probable after fatigue failure, or obvious partial failure, of a principal structural element, and that the remaining structure is able to withstand a static ultimate load factor of 75 percent of the critical limit load factor at $V_{c.}$ These loads must be multiplied by a factor of 1.15 unless the dynamic effects of failure under static load are otherwise considered.

(3) The damage tolerance evaluation of § 23.573(b).

(b) Each evaluation required by this section must—

(1) Include typical loading spectra (e.g. taxi, ground-air-ground cycles, maneuver, gust);

(2) Account for any significant effects due to the mutual influence of aerodynamic surfaces; and

(3) Consider any significant effects from propeller slipstream loading, and buffet from vortex impingements.

[Amdt. 23–7, 34 FR 13090, Aug. 13, 1969, as amended by Amdt. 23–14, 38 FR 31821, Nov. 19, 1973; Amdt. 23–34, 52 FR 1830, Jan. 15, 1987; Amdt. 23–38, 54 FR 39511, Sept. 26, 1989; Amdt. 23–45, 58 FR 42163, Aug. 6, 1993; Amdt. 23–48, 61 FR 5147, Feb. 9, 1996]

§ 23.573 Damage tolerance and fatigue evaluation of structure.

(a) *Composite airframe structure.* Composite airframe structure must be evaluated under this paragraph instead of §§ 23.571 and 23.572. The applicant must evaluate the composite airframe structure, the failure of which would result in catastrophic loss of the airplane, in each wing (including canards, tandem wings, and winglets), empennage, their carrythrough and attaching structure, moveable control surfaces and their attaching structure fuselage, and pressure cabin using the damage-tolerance criteria prescribed in paragraphs (a)(1) through (a)(4) of this section unless shown to be impractical. If the applicant establishes that damage-tolerance criteria is impractical for a particular structure, the structure must be evaluated in accordance with paragraphs (a)(1) and (a)(6) of this section. Where bonded joints are used, the structure must also be evaluated in accordance with paragraph (a)(5) of this section. The effects of material variability and environmental conditions on the strength and durability properties of the composite materials must be accounted for in the evaluations required by this section.

(1) It must be demonstrated by tests, or by analysis supported by tests, that the structure is capable of carrying ultimate load with damage up to the threshold of detectability considering the inspection procedures employed.